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(72) Inventor: John Jenkinson

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THE PATENT OFFICE
LONDON
DIEU ET MON DROIT

(54) GAS TURBINE ENGINE COOLING SYSTEM

ERRATA

SPECIFICATION NO 1531037

Page 1, line 1, (71) *delete* whole line *insert* (71) We, ROLLS-ROYCE LIMITED, a British Company of 65 Buckingham Gate, London SW1E 6AT, formerly ROLLS-ROYCE (1971) LIMITED of Norfolk House, St. James's Square, London, SW1Y 4JS, do

Page 1, lines 2 and 3, *delete* whole lines

Page 1, lines 4 and 5, *delete* whole lines

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3 April 1979

Bas 55775/18

providing a degree of cooling to the engine turbine discs and blades or for providing air seals between the turbine stages.

One of the main disadvantages with this type of system is that if cooling air is taken from the combustor section the simplest mechanical arrangement is achieved but the air temperature may be too high to permit satisfactory cooling. Alternatively, the cooling air may be taken from the high pressure compressor outer wall, however this results in an awkward and expensive structure which is necessary to convey the cooling air into the turbine. It has been found unsatisfactory to take cooling air from the high pressure compressor inner wall as large pressure losses are incurred.

An object of the present invention is to provide a gas turbine engine cooling system which incorporates the structural simplicity of bleeding air from the combustion system without suffering the temperature penalty which is usually incurred with such a system.

According to the present invention, in a gas turbine engine cooling system, cooling air is bled from the combustor section of the engine and is expanded through at least one auxiliary associated blading.

An embodiment of the invention will be described by way of example only and with reference to the accompanying drawings in which:—

Figure 1 shows a side view of a ducted fan type gas turbine engine having a partly broken away casing showing the main engine components and a diagrammatic view of an embodiment of the present invention.

Figure 2 shows an enlarged diagrammatic view of the embodiment shown diagrammatically at Figure 1.

Referring to Figure 1 of the drawings, a gas turbine engine shown generally at 10 comprises in flow series, a fan 12, a low pressure compressor 13, a high pressure compressor 14, combustion equipment 15, nozzle guide vanes 16, a high pressure turbine 17, a low pressure turbine 18, the engine terminating in an exhaust nozzle 19. The high pressure turbine and compressor 17 and 14, and the low pressure turbine and compressor 18 and 13 respectively, together with the fan 12 are each secured to the coaxially arranged high and low pressure rotatably mounted engine main shafts

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(54) GAS TURBINE ENGINE COOLING SYSTEM

(71) We, ROLLS-ROYCE LIMITED, a British Company formerly ROLLS-ROYCE (1971) LIMITED, of Norfolk House, St. James Square, London, SW1Y 4JR of 65 Buckingham Gate, London, SW1E 6AT, do hereby declare the invention, for which we pray that a patent may be granted to us, and the method by which it is to be performed, to be particularly described in and by the following statement:—

This invention relates to a gas turbine engine cooling system and more particularly to an improved turbine cooling system.

In the past it has been well known to bleed cooling air from the compressor or combustor section of a gas turbine engine. This air can subsequently be directed through the engine by means of swirl vanes or apertures to the turbine section of the engine where it is utilised in providing a degree of cooling to the engine turbine discs and blades or for providing air seals between the turbine stages.

One of the main disadvantages with this type of system is that if cooling air is taken from the combustor section the simplest mechanical arrangement is achieved but the air temperature may be too high to permit satisfactory cooling. Alternatively, the cooling air may be taken from the high pressure compressor outer wall, however this results in an awkward and expensive structure which is necessary to convey the cooling air into the turbine. It has been found unsatisfactory to take cooling air from the high pressure compressor inner wall as large pressure losses are incurred.

An object of the present invention is to provide a gas turbine engine cooling system which incorporates the structural simplicity of bleeding air from the combustion system without suffering the temperature penalty which is usually incurred with such a system.

According to the present invention, in a gas turbine engine cooling system, cooling air is bled from the combustor section of the engine and is expanded through at least one auxiliary turbine prior to being directed to the turbine section of the engine.

The at least one auxiliary turbine may comprise an impulse turbine or turbines secured to a main shaft of the engine.

In one embodiment of the invention the at least one auxiliary turbine comprises a reaction type turbine which takes the form of a plurality of angled holes disposed within a hollow main shaft of the engine.

Preferably the hollow main shaft comprises the high pressure engine shaft.

The cooling air is expanded through the at least one auxiliary turbine to reduce both its pressure and temperature prior to being directed to the turbine section of the engine.

Preferably the cooling air is used to cool the high pressure turbine disc and may also be used to cool the high pressure turbine discs associated blading.

An embodiment of the invention will be described by way of example only and with reference to the accompanying drawings in which:—

Figure 1 shows a side view of a ducted fan type gas turbine engine having a partly broken away casing showing the main engine components and a diagrammatic view of an embodiment of the present invention.

Figure 2 shows an enlarged diagrammatic view of the embodiment shown diagrammatically at Figure 1.

Referring to Figure 1 of the drawings, a gas turbine engine shown generally at 10 comprises in flow series, a fan 12, a low pressure compressor 13, a high pressure compressor 14, combustion equipment 15, nozzle guide vanes 16, a high pressure turbine 17, a low pressure turbine 18, the engine terminating in an exhaust nozzle 19. The high pressure turbine and compressor 17 and 14, and the low pressure turbine and compressor 18 and 13 respectively, together with the fan 12 are each secured to the coaxially arranged high and low pressure rotatably mounted engine main shafts

SEE ERRATA SLIP ATTACHED

20 and 21, and the fan 13 is arranged to rotate within an annular fan duct 22 which surrounds the remainder of the engine. A cooling system made in accordance with an embodiment of this invention is shown merely diagrammatically in this view in direction of arrow 23.

Figure 2 of the drawings shows on an enlarged scale a portion of the engine shown diagrammatically at Figure 1 comprising a portion of the flame tube 24 and the dilution or secondary air casing 25 which forms a part of the combustion equipment 15 from which cooling air is bled through a circumferentially extending aperture 26.

The circumferentially extending aperture 26 is arranged such that the cooling air issuing from it when the engine is in operation is directed to impinge upon an impulse turbine, one blade of which is shown at 27. The impulse turbine 27 is secured to the high pressure engine shaft 20 and therefore augments the drive of the high pressure turbine 17 which is attached to the same shaft. The air passing through the impulse turbine 27 will therefore suffer both a pressure and a temperature reduction due to providing work to the high pressure shaft 20.

In this embodiment of the invention it has been stated that the auxiliary turbine 27 is of the impulse type. This has been chosen merely as a matter of convenience and simplicity. The impulse turbine could be substituted for a reaction type turbine, however, this would necessitate the use of a turbine shroud or some other apparatus to satisfactorily overcome the sealing problems.

The cooling air after being expanded and cooled by means of the auxiliary turbine 27 passes downstream through the engine and a portion of it passes through apertures 32 and 33. A portion of this air then passes through the seal 34 into the main gas stream of the engine and the remainder is directed into the turbine blade root 30 to be used for blade cooling purposes. The remainder of the cooling air is subjected to further expansion and cooling by passing through a further auxiliary turbine 28 which is also secured to the high pressure shaft 20 and provides a further drive to it. A portion of the further expanded and cooled air passes up the front face of the turbine disc 29 and then passes into the blade root 30 thus augmenting the flow of cooling air for the turbine blades 17. The remainder of the further cooled air passes through a plurality of angled holes situated in the high pressure engine shaft 20, one of the holes being shown at 31. The holes 31 are arranged such that their axes are arranged at an angle to the radial axis of the high pressure engine shaft 20 within which they are situated such that they act as a reaction turbine on the cooling air passing through them. The holes 31 therefore subject the cooling air passing through them to a further expansion and cooling process and provide further drive to the high pressure shaft 20 within which they are situated. The further cooled air subsequently passes around the disc 28 to provide it with a degree of cooling before being exhausted into the main gas stream of the engine through the interstage turbine seals (not shown in the drawings).

It is believed that by use of this invention the temperature of the cooling air is substantially reduced, this can result in two main advantages in that it will permit the turbine disc and blades to be manufactured from a less sophisticated metal than would usually be necessary using a conventional cooling system, therefore leading to a significant cost saving on material. Alternatively the invention would permit the use of increased compressor operating temperatures which gives the advantage of increased engine efficiency, performance and flight speed.

WHAT WE CLAIM IS:–

1. A gas turbine engine cooling system in which cooling air is bled from the combustion section of the engine and is expanded through at least one auxiliary turbine prior to being directed to the turbine section of the engine.

2. A gas turbine engine cooling system as claimed in claim 1 in which at least one auxiliary turbine comprises an impulse turbine or turbines secured to a main shaft of the engine.

3. A gas turbine engine cooling system as claimed in claim 1 in which the at least one auxiliary turbine comprises a reaction type turbine which takes the form of a plurality of angled holes disposed within a hollow main shaft of the engine.

4. A gas turbine engine cooling system as claimed in claim 3 in which the hollow main shaft of the engine comprises the high pressure engine shaft.

5. A gas turbine engine cooling system as claimed in any preceding claim in which the cooling air is expanded through the at least one auxiliary turbine to reduce both its pressure and temperature prior to being directed to the turbine section of the engine.

6. A gas turbine engine cooling system as claimed in any preceding claim in which the cooling air is used to cool the high pressure turbine disc.

7. A gas turbine engine cooling system as claimed in any preceding claim in which the cooling air is used to cool both the high pressure turbine disc and associated turbine blading.

8. A gas turbine engine cooling system substantially as herein described and as illustrated in the accompanying drawings.

J. WAITE,
Chartered Patent Agent,
For the Applicants.

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1531037 COMPLETE SPECIFICATION

1 SHEET *This drawing is a reproduction of the Original on a reduced scale*

Fig.1.

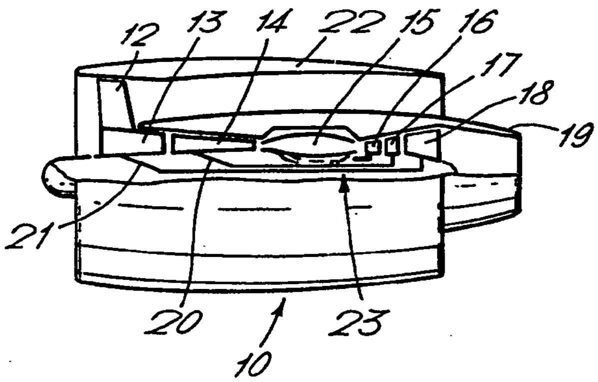

Fig.2.